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RESEARCH MEMORANDUM

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By Morris W. Rubesin
Ames Aeronautical Laboratory

Charles B. Rumsey
Langley Aeronautical Laboratory

and Steven A. Varga
Ames Aeronautical Laboratory

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A SUMMARY OF AVAILABLE KNOWLEDGE CONCERNING SKIN FRICTION AND HEAT TRANSFER AND ITS APPLICATION TO THE DESIGN OF HIGH-SPEED MISSILES[1]

By Morris W. Rubesin
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To determine the skin friction and heat transfer on the surfaces of high-speed missiles, it is necessary to know certain characteristics of the boundary layers. These characteristics are: the temperature recovery, the skin-friction coefficients and the heat-transfer coefficients of both the laminar and turbulent boundary layers and, also, the position of the transition from laminar to turbulent flow. In this paper a review is made of the existing information concerning these characteristics. In addition, comparison is made between existing flight data and results computed by the boundary-layer momentum-integral method in a preliminary attempt to establish some rational way of approaching the design of a missile whose Mach number range and body geometry are markedly different than those of existing data.

The problem of determining the position of transition from a laminar to a turbulent boundary layer is very important. For flight conditions in which transition occurs in regions other than very near the nose of the missile, the average skin friction and heat transfer may be influenced more by the location of transition than by the absolute values of the skin friction or heat transfer corresponding to either the laminar or turbulent flow. At present there is no accurate method for determining the location of transition.

A compilation of data showing the beginning and end of boundary-layer transition is shown in figure 1. The ordinate is the Reynolds number based on the length along the body. The abscissa is the free-stream Mach number. Open symbols designate the beginning of transition, whereas the filled-in symbols designate the end of transition. Most of these data, compiled by project Hermes, were obtained in the early stages of V-2 flight and are, in effect, for a cooled surface.

[1]This is substantially a reprint of the paper by the same authors which was presented at the NACA Conference on Aerodynamic Design Problems of Super sonic Guided Missiles at the Ames Aeronautical Laboratory on Oct. 2-3, 1951.

Also shown are data obtained on unheated flat plates at Princeton (reference 1) and in the Ames 6-inch heat transfer tunnel (reference 2), on an unheated RM-10 test body in the Langley 4- by 4-foot supersonic tunnel, and on an unheated body of revolution at the Lewis Laboratory. It can be seen that the scatter in the data is enormous; however, the general trend of the data indicates that the Reynolds numbers of the beginning and end of transition increase at the higher Mach numbers. That the data shown do not correlate any better is expected. In these data no control was made of such important quantities as surface roughness, body shape, free-stream turbulence, and surface temperature.

One of these variables, the surface temperature, was isolated for study in tests performed at the Ames Laboratory on a heated flat plate at $M = 2.4$ (reference 2). The results of these tests are shown in figure 2. The ordinate used is the Reynolds number based on the momentum thickness whereas the abscissa is the ratio of the surface temperature to the free-stream temperature. This form of Reynolds number was chosen to localize conditions, thereby making the results applicable to bodies of revolution with surface-pressure variations. The points at the extreme left are for the unheated case. It can be seen that the Reynolds numbers of the beginning and end of transition are reduced by about 50 percent from the unheated condition for a surface to free-stream temperature ratio of 2.8. It is interesting to note that a determination of the momentum thickness Reynolds number at the beginning of transition on an unheated body of revolution tested at the Lewis Laboratory resulted in a value identical to that shown for the unheated plate in this figure. This agreement may have been fortuitous because the Mach number of the body was 3.12 whereas that for the plate was 2.4.

Obviously, much more work needs to be done concerning transition before an accurate means is available for predicting its position on a missile. Because there is no alternative, it is recommended that until more information is available the results of figure 1 be used as a guide in design.

Before it is possible to determine the heat transfer, and often the skin friction, it is necessary to know the recovery temperature. The recovery temperature can be determined from the usual equation for recovery factor shown in figure 3. In the equation at the left r is the recovery factor, T_r is the recovery temperature of an unheated body, T_∞ is the free-stream temperature, and M_∞ is the free-stream Mach number. This figure shows a compilation of temperature recovery factor as a function of Mach number obtained in wind tunnels at Mach numbers below 4, and for two flight tests at Mach numbers approximately equal to 2. These data apply to flat plates and bodies of revolution. The length of the vertical bars, which represent wind-tunnel data, shows the range of variation of the recovery factor with Reynolds number at a fixed Mach number. In general, the data points lie on two levels, around $r = 0.85$ and $r = 0.90$. These levels agree with the usual theoretical values of $Pr^{1/2}$ for laminar flow and $Pr^{1/3}$ for turbulent flow. The set of data for the laminar boundary layer on a flat plate is not in agreement with the other laminar-boundary-layer data, or the theoretical prediction.

In addition, two theoretical results showing the effect of Mach number on the recovery factor are also indicated. The turbulent-boundary-layer recovery factor determined by Tucker and Maslen (reference 3) by extending the approximate Squire analysis to include compressibility shows a reduction with Mach number. Apparently the variation of the recovery-factor data does not exhibit this change. It can be concluded, therefore, that the usual theoretical value $Pr^{1/3}$ be used for the recovery factor in design work through the Mach number range, neglecting the theoretical variation indicated. For the laminar boundary layer, Klunker and McLean (reference 4) have shown that, under flight conditions where extremely high air temperatures occur, the recovery factor decreases with Mach number. These results were obtained from the same basic boundary-layer theory which yields a recovery factor of $Pr^{1/2}$ for the temperature levels occurring in wind tunnels. Thus, the agreement of wind-tunnel data with $Pr^{1/2}$ checks the basic theory. The flight datum point shown is at too low a Mach number to indicate any marked reduction. Since the experimental data agree with the basic theory, the work of Klunker and McLean for flight conditions should yield satisfactory results for design purposes at high Mach numbers.

Several theories exist for determining the magnitude of the skin-friction and heat-transfer coefficients. For the case of flight conditions where extremely high temperatures occur, the previously mentioned theory of Klunker and McLean also provides a means to calculate the laminar-boundary-layer skin-friction and heat-transfer coefficients. In addition, Van Driest (reference 5) has obtained similar results by extending the work of Crocco to include flight conditions with the resulting high air temperatures. Although the Crocco method is restricted in that the Prandtl number is assumed constant and the viscosity is expressed in Sutherland's equation in terms of enthalpy rather than temperature, figure 4 indicates that the results of average skin friction for the laminar boundary layer at Mach numbers below 10 are within 1 percent of the more exact method of Klunker and McLean. As good agreement is also obtained for the recovery temperature and the local heat transfer, it can be concluded that for practical purposes the two theories give equal results.

The data with which these theories can be compared are relatively meager. Published skin-friction data on unheated flat plates (references 2, 6, and 7) represent the average skin friction from the leading edge to the point of measurement of boundary-layer surveys. These average skin-friction coefficients obtained at Mach numbers around 2 are about 30 percent higher than those given by Crocco's theory made to apply to wind-tunnel conditions. Similar results were obtained at Lewis from unpublished data on a hollow cylinder placed parallel to the air stream. This discrepancy between theory and experiment has been attributed to the momentum loss in the boundary layer caused by the bluntness of the sharp leading edge. Unpublished data of average skin friction obtained at the Langley Laboratory on a 6° wedge in a flow at a Mach number of 6.9 exceeds by about 14 percent the estimated theoretical value based on the Crocco method when the wedge is at a zero angle of attack. Further unpublished tests at the Lewis Laboratory have indicated that the laminar-boundary-layer theories compare favorably with the experimental average skin-friction coefficients determined experimentally on a cone-cylinder body at $M = 3.85$. Although no local skin-friction data have as yet been correlated with the theory, local heat-transfer data shown in figure 5 have been determined on a cone having approximately a constant surface temperature (reference 8). The data are, on the average, about 12 percent lower than those given by the Crocco theory, corrected to a cone. In general, it can be concluded that the Crocco theory predicted the skin friction and heat transfer within engineering accuracy up to a Mach number of 7, for the wind-tunnel tests. It then would be expected that the theories for the laminar boundary layer for flight conditions are adequate for design.

For the turbulent boundary layer there are several theories from which the skin friction and the heat transfer on flat plates can be calculated (reference 9). Each of these theories indicates a marked reduction in the skin-friction and heat-transfer coefficients with an increase in Mach number or surface temperature. Because of the large effects indicated by the theories and because they are of a semiempirical nature it is important to compare them with existing data. This comparison is made in figure 6 for the case of an unheated flat plate in a wind tunnel at a Mach number of 2.4 (reference 9). It is observed that the average skin-friction coefficient is reduced from the values of the incompressible case; however, the reduction estimated by Von Kármán was not realized. The compressible theories for turbulent flow on a flat plate give good agreement with the data over the range of Reynolds numbers below 6,000,000. In figure 7 are shown unpublished local skin-friction data obtained on unheated cylinders with their axes placed parallel to the air flow. The Mach number of these tests was 3.1. The abscissa used in this figure is the Reynolds number based on the momentum thickness. This characteristic dimension was used to avoid the necessity of knowing the exact location of transition. In the lower figure there are shown the data with natural transition. These data agree approximately with the compressible flat-plate theories at the lower Reynolds numbers. Beyond a Reynolds number of 6000 the data drop off toward the Von Kármán estimation. The data with artificial transition shown in the upper figure exhibit a reduction from the incompressible case; however, the data have a different slope than any of the theories and give no insight into which of the theories agree best with the physical phenomena.

Figure 8 is intended to show that a modified Reynolds analogy exists at a Mach number of 2.4. This unpublished datum point was obtained on a cooled flat plate in the Ames 6-inch heat transfer tunnel. The ordinate is written in a fashion which permits comparing heat-transfer data with theoretical skin-friction computations through a modified Reynolds analogy. The abscissa used in this figure is the Reynolds number based on the momentum thickness to avoid the necessity of knowing the location of transition. The single datum point of heat transfer compares favorably with the theories of Frankl and Voishel and of Van Driest.

In general, it can be concluded from the last three figures of wind-tunnel data that the compressible-turbulent-boundary-layer theories represent the available data of skin friction and heat transfer on flat plates with an accuracy sufficient for design. The same cannot be said from the data obtained on cylinders with their axes parallel to the air stream, except for the data obtained with natural transition at Reynolds numbers below 6000 when based on momentum thickness which did agree fairly well with the theories.

Data of skin friction and heat transfer have been measured in flight on the RM-10 missile, the earliest of which are included in references 10 and 11. Figure 9 shows time histories of the flight characteristics for a typical boosted RM-10 flight during which average skin-friction coefficients were obtained from boundary-layer rake measurements to a maximum Mach number of 3.7. The characteristics shown are a surface temperature parameter, the Reynolds number based on the length to the rake location just ahead of the fins, the Mach number, and the average skin-friction coefficient.

The surface temperature parameter shown was used since its numerical value indicates the magnitude of the heating regardless of Mach number and indicates cooling and heating of the boundary layer by negative and positive values, respectively. The experimental skin-friction coefficients are from 20 to 30 percent higher than Van Driest's theoretical prediction for a flat plate at the test conditions, except near peak Mach number. During the first part of the test which is after booster separation but prior to firing of the sustainer rocket, transition would be well forward on the pointed nose of the missile so that close to 100 percent of the skin area would have turbulent boundary layer and the measured values would be average turbulent coefficients. It is expected, however, that at the high Mach numbers during sustainer firing, the strong cooling of the boundary layer indicated by the surface temperature parameter would stabilize the laminar boundary layer and cause transition to move back on the body. The measurements during this part of the flight would thus be lower than average turbulent coefficients. During the period after sustainer firing, the heating parameter became less stabilizing, and transition would be expected to move forward causing a relative rise in the average coefficient. These trends are shown by the data.

At a time of about 23 seconds, the heating parameter became positive, or destabilizing, and nearly all of the skin area would again be covered by turbulent flow. The 20 to 30 percent difference shown between turbulent flat-plate theory and the data for times of nearly complete turbulent boundary layer is attributed to the missile geometry and to the pressure distribution at the flight Mach numbers.

Figure 10 shows unpublished flight conditions and results from a cylindrical body with an ogive nose. This configuration more closely approximates a flat plate. The measured values of average skin-friction coefficient are relatively lower than the RM-10 results and are in close agreement with Van Driest's flat-plate theory. The extent of laminar flow on this model is believed to have been small because of the values of Mach number and Reynolds number, at least during the first half of the test.

Presented in figure 11 are values of average skin-friction coefficient at the condition of zero heat transfer which have been obtained at four points in the skin-friction tests, all occurring at a Reynolds number of about 60×10^6 but at different Mach numbers from 1.1 to 3. Also shown is a value at zero Mach number and 60×10^6 Reynolds number which was recently obtained from rake measurements in under-water tests performed on an RM-10 body in the Langley tank no. 1. A flat-plate theory is also included to show its variation with Mach number. Below Mach number 1 no reduction is shown by the data. From Mach number 1 to 3, the data show a reduction of about 30 percent whereas the flat-plate theory shows a reduction of 35 percent.

Local heat-transfer coefficients measured in flight on the RM-10 missile are shown in figure 12. The data are plotted as $NuPr^{-1/3}$ against Reynolds number with the velocity and air properties based on free-stream conditions. Above a Reynolds number of about 6×10^6 the heat-transfer coefficients are for turbulent flow. Below approximately 2×10^6, the coefficients measured on the nose of model C show a decrease from the turbulent correlation indicative of laminar flow. The values are, however, considerably higher than the laminar theory for a cone. Plotted in the present manner the data lie midway between the laminar-boundary-layer theory for a cone and the measured turbulent data.

The heat-transfer data for turbulent boundary layer can be represented to ±7 percent by a line having the equation indicated. These data were obtained over a Mach number range from 1 to 2.8 and at several stations along the body as indicated in the legend. It is interesting to note that, for three models, almost all of the data for all stations along the length of the body and over the complete Mach number range agree to within ±7 percent.

It is concluded from the flight-test data that for missiles not greatly different in shape from the RM-10, and for conditions similar to the test range, heat-transfer characteristics for turbulent flow can be obtained from the RM-10 equation for design purposes. It should be emphasized that the heat-transfer data do not explicitly show a Mach number effect in the range of Mach numbers tested. The test data further indicate that the skin-friction coefficients can be obtained by reference to the flat-plate theory in the following manner. For ogive-cylinder bodies practically no modification to the theory is necessary. For bodies of higher fineness ratio than the RM-10 it would be expected that the values of skin friction are between those of the RM-10 and the flat plate.

In view of the conclusions drawn from the flight-test data, it is apparent that some rational method is necessary for extrapolating the known data to blunter bodies or to bodies flying at flight conditions much different than those of the available tests. As the flat-plate theories including compressibility agreed well with the skin-friction data obtained with the ogive cylinder, it was believed that some method accounting for body shape might bring the theories in line with the data obtained on the RM-10, thereby extending the scope of the data. Therefore, computations were made of heat transfer and skin friction for the RM-10 shape and flight conditions by means of the well-known momentum-integral method using the Frankl and Voishel flat-plate theory.

The momentum-integral method consisted of solving the equation shown in figure 13. This equation relates the rate of growth of the boundary layer with the compressibility effect, the acceleration of the air outside the boundary layer, the geometry of the body, and the local skin-friction coefficient. The solution of this equation is obtained through the use of the flat-plate relationships of the skin-friction coefficient and the Reynolds number based on the momentum thickness. The solution yields the distribution of the momentum thickness, from which the skin-friction coefficient can be determined. The local heat-transfer coefficient is obtained from the local skin-friction coefficient through a modified Reynolds analogy.

In figure 14 there is shown a comparison of some preliminary results of the momentum-integral method with the local heat-transfer coefficients measured on the RM-10. For the theoretical computations the local skin-friction coefficient was expressed in terms of the Reynolds numbers based on momentum thickness according to the flat-plate theory of Frankl and Voishel. The ordinate shown is the local heat-transfer coefficient. The abscissa is the dimensionless length along the body. Two sets of data are shown; the upper set is for a Mach number of 2.3, whereas the lower is for a Mach number of 1.02. It should be noted that the Reynolds numbers of these data are roughly in proportion to the Mach numbers. The solid lines represent the distributions given by the equation representing the bulk of the RM-10 data. The dashed line represents the results obtained from the momentum-integral method. At the lower Mach number, and consequently the lower Reynolds number, the momentum-integral method agrees well with the data and the RM-10 equation. The results of the Van Driest flat-plate theory for these conditions were about 10 percent lower than the data along the entire body. At the higher Mach number the momentum-integral method gave results which are about 15 percent higher than the data on the front of the missile and about 3 percent higher than the data towards the rear of the missile. The data apparently do not show the geometry effect expected from the momentum-integral method. In fact, the Van Driest flat-plate theory gives results which pass through the data near the front of the missile and then drop to values about 3 percent low in the rear portions of the missile. From the latter results it can be concluded that, for slender bodies such as the RM-10, the RM-10 equation or flat-plate theory represents the data as well as does the more tedious momentum-integral method at a Mach number of 2.3. The momentum-integral method may become necessary for blunter bodies.

The momentum-integral method is evaluated further in figure 15. The ordinate shown is the average skin-friction coefficient and the analogous heat-transfer parameter. The abscissa is the Mach number. The average skin-friction data shown are for the RM-10 at recovery temperature. These data were shown previously in figure 11. The average heat-transfer parameter shown was evaluated from the RM-10 equation and it is noted there is no Mach number effect in this equation representing the bulk of the RM-10 data. For comparison, two flat-plate theories corresponding to the recovery temperature are included. These theories are for conditions comparable with the curve of the RM-10 heat-transfer parameter because the same value of the parameter was obtained under conditions of both cooling and heating.

It is noted that the momentum-integral method does not reconcile the flat-plate theories with the characteristics of the RM-10 data in that first, the momentum-integral method does not sufficiently increase the values obtained using flat-plate theory to agree with the RM-10 skin-friction data, and second, the Mach number effect remains in the theories even when based on the momentum-integral method thereby resulting in a lack of agreement with the RM-10 heat-transfer equation. It is apparent that no conclusive method for extrapolating existing data to greatly different conditions can be given at present.

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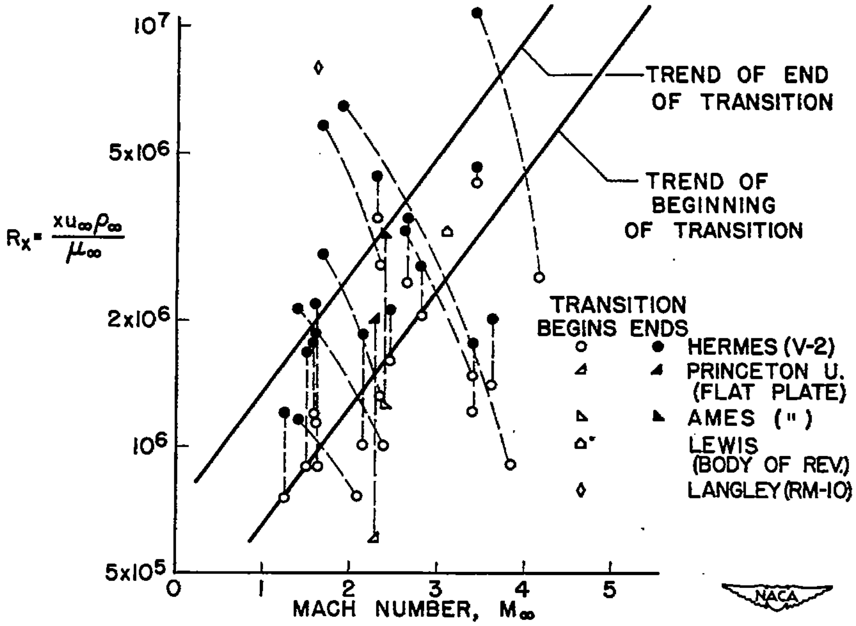

Figure 1.- Location of boundary-layer transition on different bodies in flight and in wind tunnels.

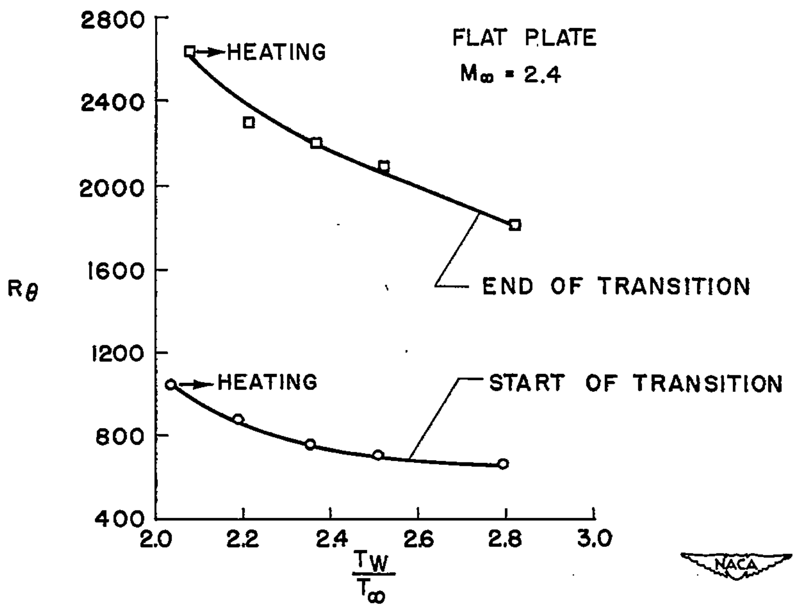

Figure 2.- Effect of heating on boundary-layer transition.

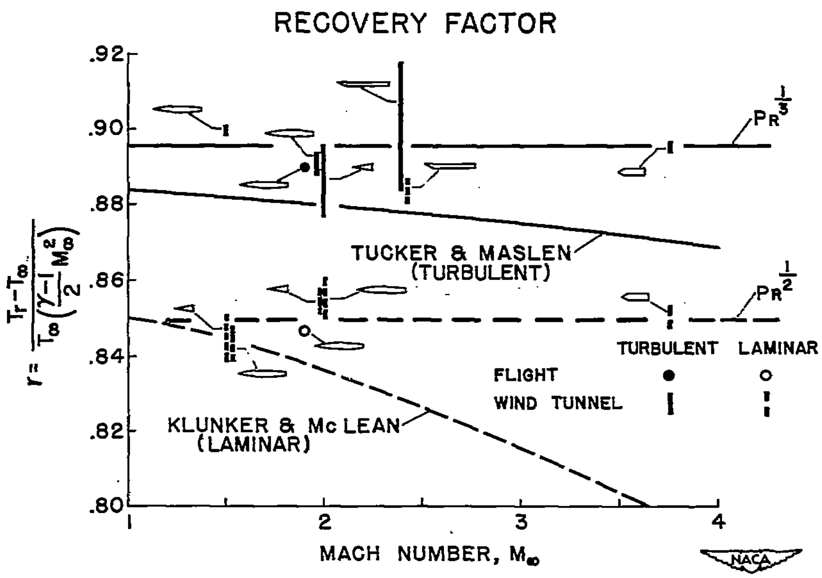

Figure 3.- Compilation of temperature recovery factors and comparison with theories.

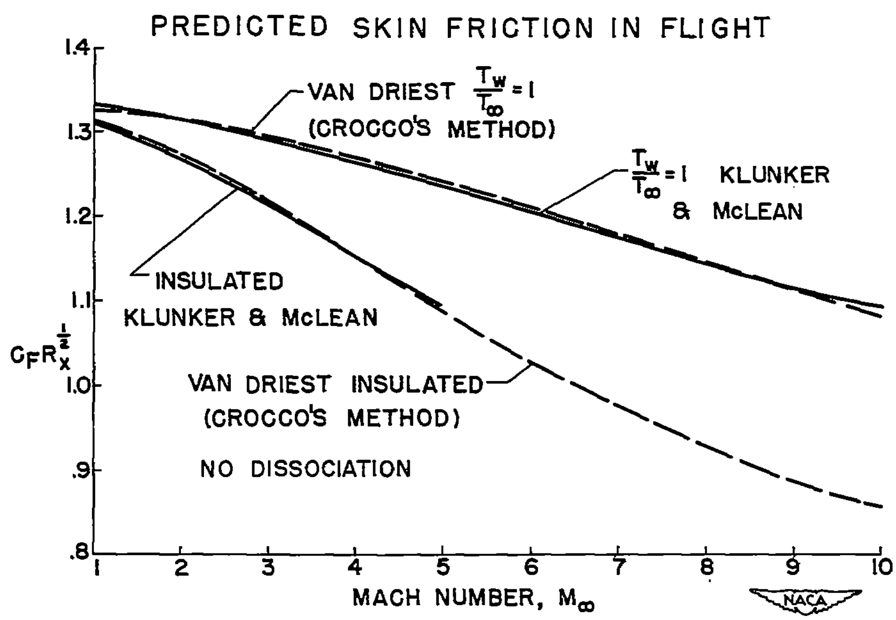

Figure 4.- Comparison of the theories of Van Driest (Crocco) and of Klunker and McLean for flight conditions.

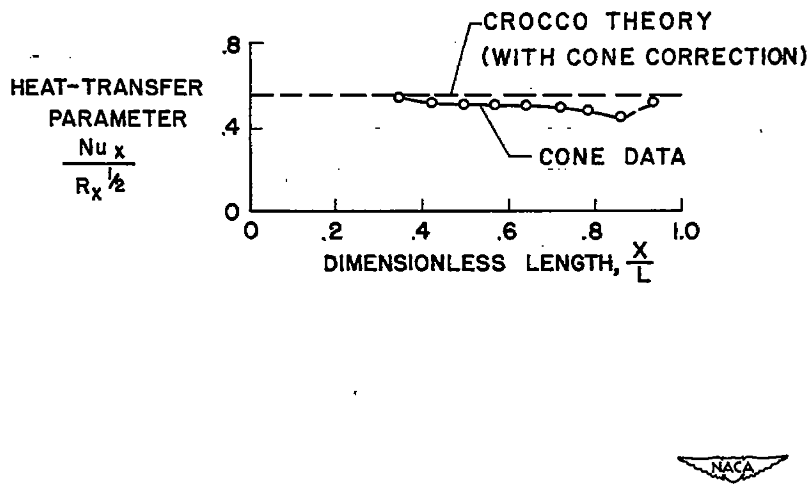

Figure 5.- Comparison of Scherrer's cone data with Crocco's theory.

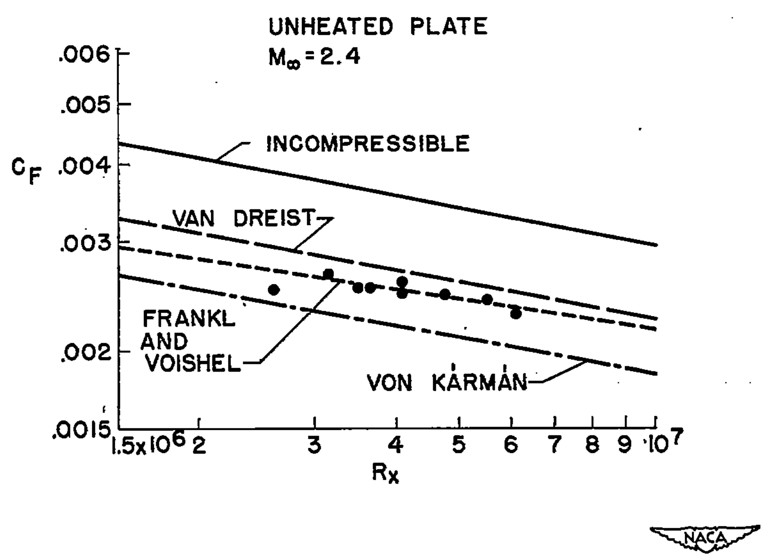

Figure 6.- Comparison with theories of experimental average skin-friction coefficients on a flat plate.

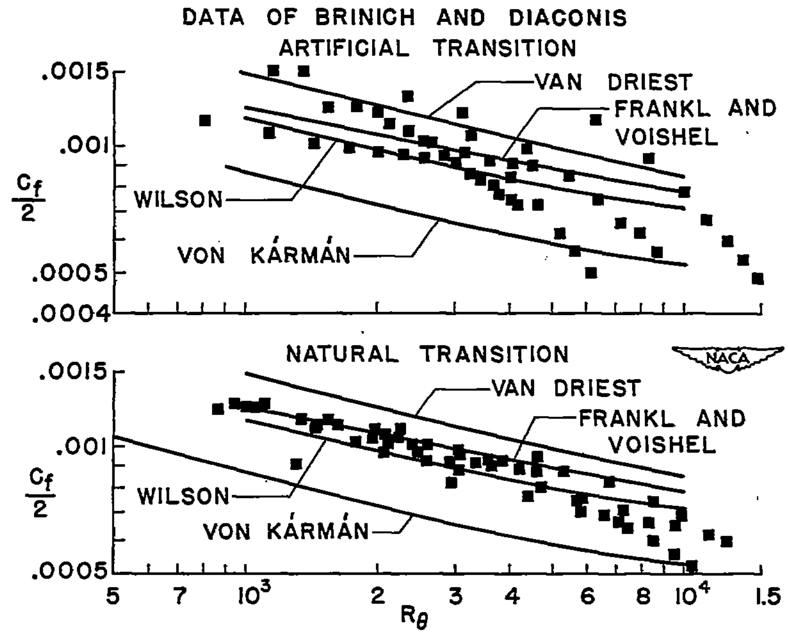

Figure 7.- Comparison with theories of experimental average skin-friction coefficients on a cylinder in axial flow.

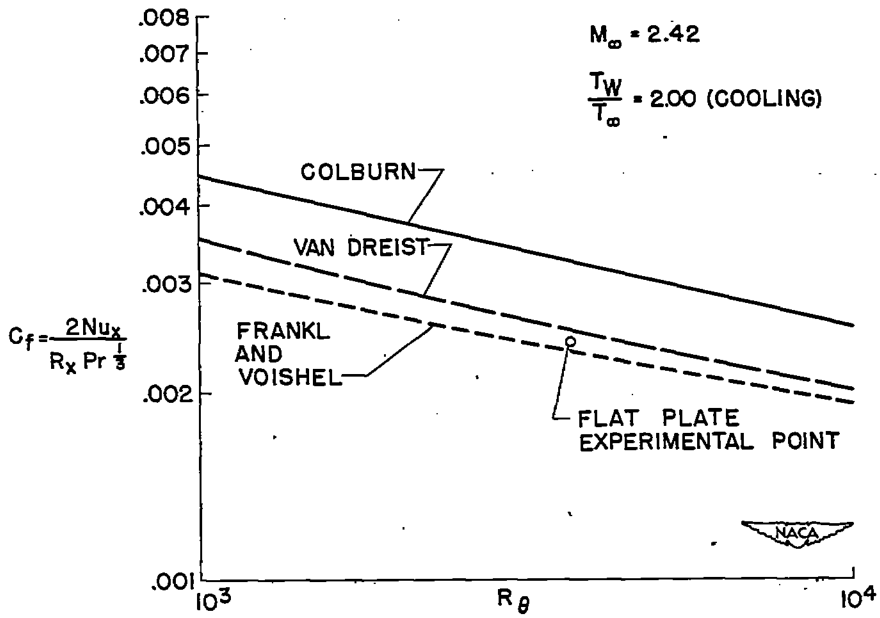

Figure 8.- Comparison with theories of experimental local heat-transfer coefficient on a flat plate.

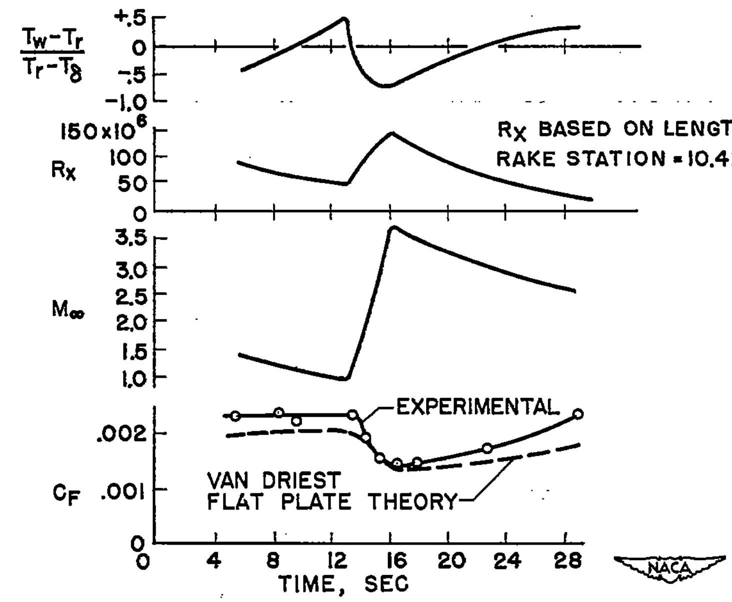

Figure 9.- Flight data from the RM-10 missile and comparison of experimental average skin-friction coefficients with theory.

TIME HISTORIES OF OGIVE-CYLINDER BODY TEST

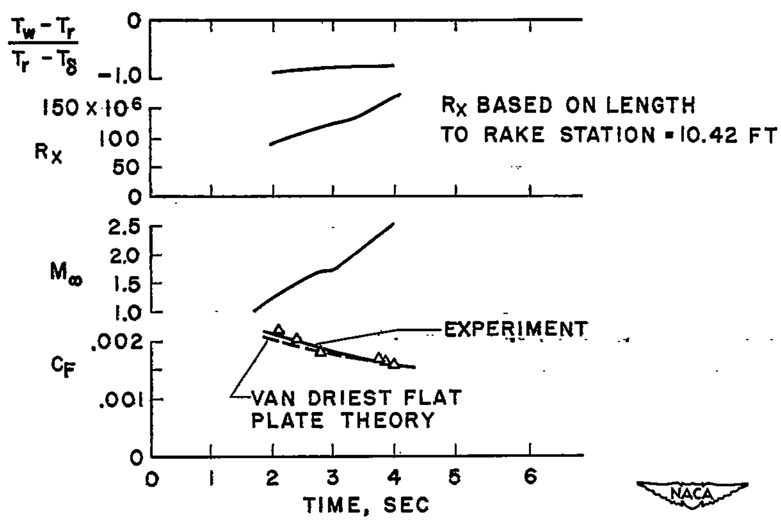

Figure 10.- Flight data from ogive-cylinder body and comparison of average skin-friction coefficients with theory.

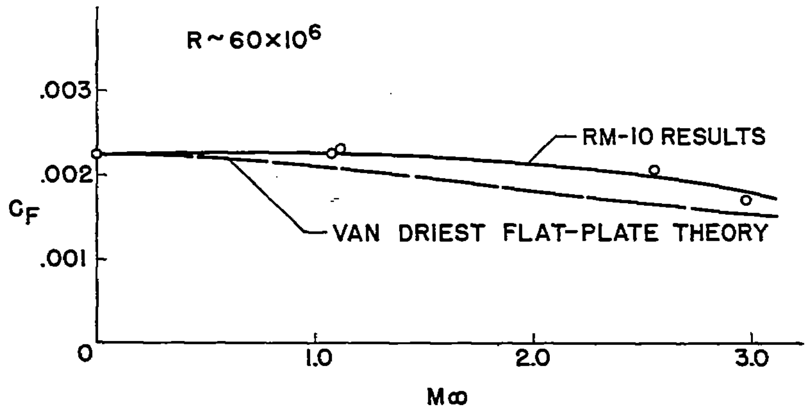

Figure 11.- Comparison with theory of experimental average skin-friction coefficients on the RM-10 in flight.

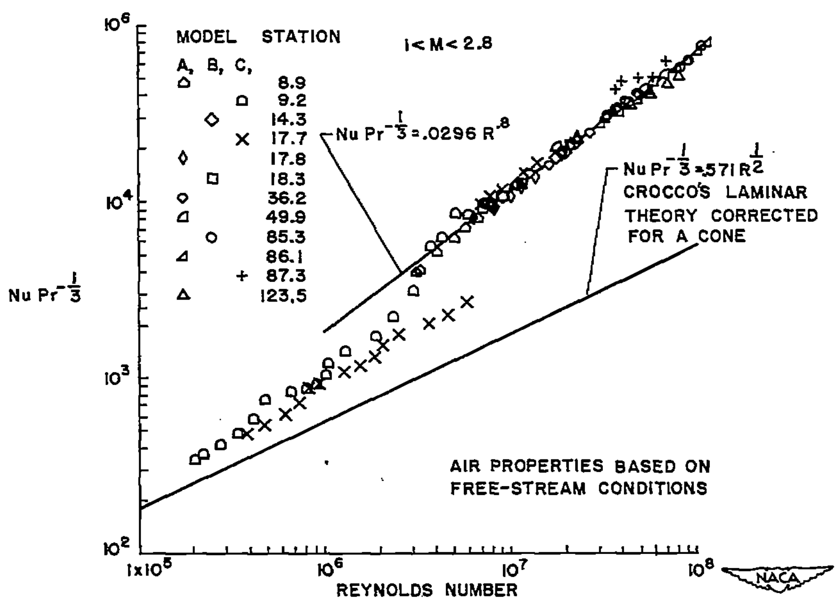

Figure 12.- Heat-transfer results from RM-10 flight tests.

$$\frac{d\theta}{dx}+\left\{\frac{H+2-M_1^2}{M_1\left(1+\frac{\gamma-1}{2}M_1^2\right)}\frac{dM_1}{dx}+\frac{1}{r}\frac{dr}{dx}\right\}\theta=\frac{C_f}{2}$$

Figure 13.- Integral-momentum equation for bodies of revolution in compressible flow.

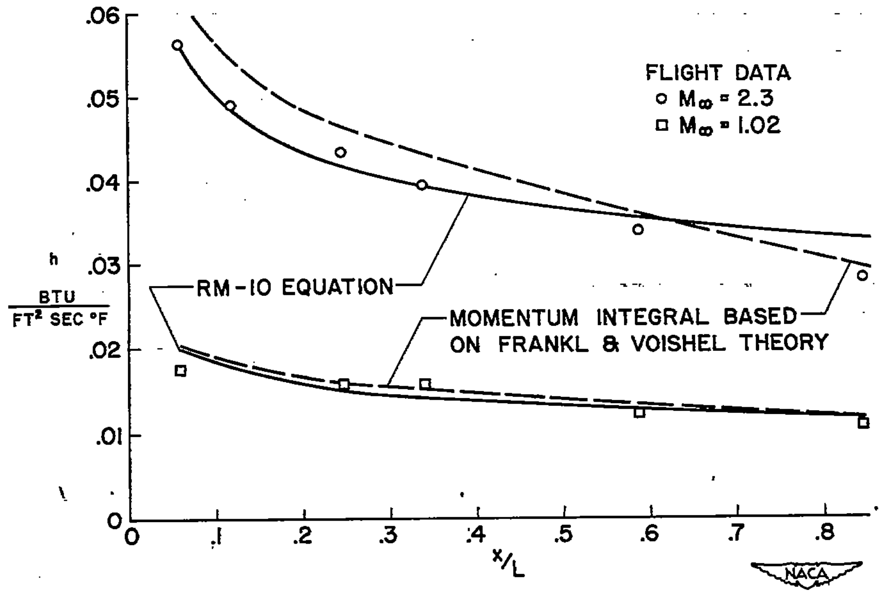

Figure 14.- Comparison of experimental and theoretical heat-transfer coefficients on RM-10.

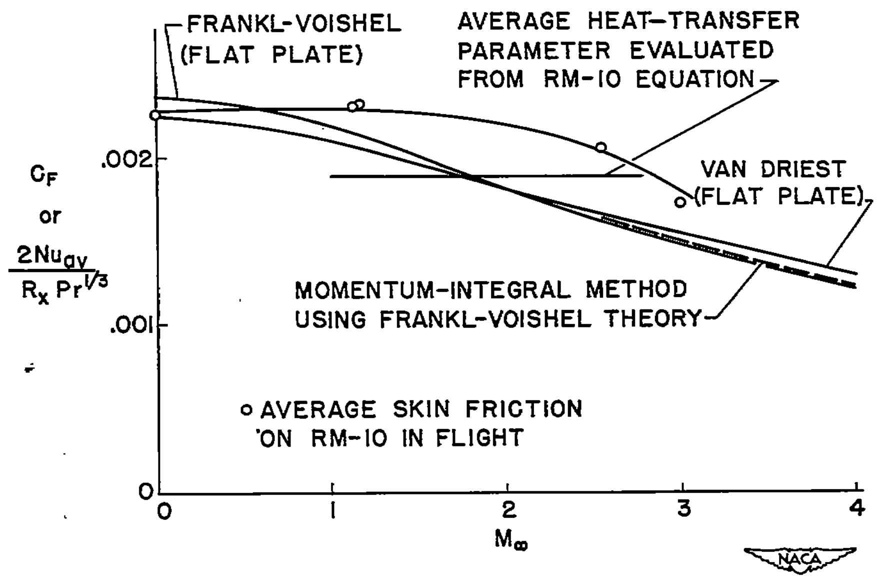

Figure 15.- Comparison with theories of average skin-friction coefficients on RM-10 in flight.

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